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Code 5

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Technical Report No. 32-26

Some Considerations for the Selection of Upper-Stage Propellants

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N-94346

index only

JPL TR 32-26

(NASA CR-50766)

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CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

April 5, 1960

RQT-7576

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
(NASA CONTRACT ~~NO.~~ 2 NASw-6)

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Some Considerations for the Selection of Upper-Stage Propellants

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April 5, 1960

CR - 50,766

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ABSTRACT

A comparison is presented of the characteristics of several propellant combinations which influence their selection for use in an upper-stage vehicle application. For equal cost and reliability, stage performance, as given by the ratio of payload weight to gross weight, is a measure of the desirability of propellant combinations. Superior performance can be expected from the fluorine–hydrogen propellant combination in all pumped systems. The fluorine–hydrazine combination is probably the most versatile for pressurized-system applications. The oxygen–hydrogen propellant combination can be expected to give a performance which closely approximates that of fluorine–hydrogen in pumped systems; however, oxygen–hydrogen is inferior to both fluorine–hydrogen and fluorine–hydrazine for pressurized-system applications. The combinations oxygen—RP-1 and nitrogen tetroxide–hydrazine give considerably poorer performance in both pumped and pressurized systems.

The relatively low cost of the development of a new vehicle for upper-stage application as compared to the total cost of the space program emphasizes the importance of propulsion-system performance and reliability. Inherent propulsion-system reliability might be adversely affected by various propellant properties. Only the most severe requirements for ultimate reliability or short lead time would justify the selection of low-performance propellants for a new upper-stage application.

I. INTRODUCTION

The selection of the propellant combination which will result in the maximum performance of an upper stage is dependent on many factors. As a rule, the selection is made for a particular vehicle and mission (Ref. 1 and 2); however, it is possible to determine those general characteristics of propellant combinations which influence their selection in any application. Some of the general characteristics are presented and the general applicability of several propellant combinations is discussed in the following sections.

II. DISCUSSION

For a comparison of propellant characteristics, five combinations of propellants were chosen. It was assumed that all engines would be operated in a vacuum; therefore, the chamber pressure (P_c) for the pump-fed engines was selected as 400 psi, with an expansion ratio of 30:1. The pressure-fed engines had a chamber pressure of 150 psi and an expansion ratio of 20:1. It is believed that these values are typical of the requirements for an upper-stage application. It was also assumed that the combustion efficiency and the turbine gas energy (for pump-fed engines) would increase slightly with the more energetic propellants. For all cases, a nozzle-loss factor of 0.965 was used. The mixture ratio used in all combinations was slightly less than that for maximum shifting equilibrium performance. The effects of these assumptions and the calculated effective exhaust velocity (c) are given in Tables 1 and 2. With these values for the effective exhaust velocity, the parametric study of stage performance was made using the familiar Equation

$$\Delta V = c\, ln\left(\frac{W_g}{W_g - W_p}\right)$$

where

ΔV = stage gross (or field-free) velocity

c = engine effective exhaust velocity

W_g = stage gross weight at launch

W_p = stage useful-propellant weight

in the form

$$\frac{W_{pl}}{W_g} = 1 - \frac{1}{\nu_p}\left[1 - \frac{1}{e^{(\Delta V/c)}}\right]$$

where

W_{pl} = stage payload weight (including guidance)

ν_p = propellant fraction at launch, $W_p/(W_g - W_{pl})$

The use of this Equation implies that there is no difference in velocity losses due to gravity among the systems with different propellants, which is reasonable for upper stages if the initial thrust-to-weight ratio is sufficiently high ($\geqq$0.5). Figures 1 through 4 show the relation of the payload fraction to propellant fraction for the different propellant combinations. The Figures represent two different values of gross velocity for the pumped and pressurized systems. For purposes of comparison, a two-stage N_2O_4–N_2H_4 system is shown on the high gross-velocity Figures (2 and 4). The data points on the Figures represent typical system designs which will be discussed subsequently. There is considerable displacement, both vertical and lateral, among the curves representing the different propellants. As can be seen, the lateral displacement is reduced with increasing gross-velocity requirements, which demonstrates the importance of high propellant fractions in high gross-velocity missions. By a comparison of high and low gross-velocity curves (Fig. 1 and 3 with 2 and 4), it can be seen that the effect of increasing specific impulse becomes greater as ΔV increases.

Table 1. Characteristics of pumped systems

Propellants	Turbopump factor	Combustion efficiency	Mixture ratio	Bulk density, g/cc	Effective exhaust velocity, ft/sec
N_2O_4–N_2H_4	0.985	0.965	1.25	1.22	9,950
O_2–RP-1	0.98	0.955	2.33	0.997	10,150
F_2–N_2H_4	0.985	0.965	2.2	1.32	12,350
O_2–H_2	0.99	0.97	5.0	0.325	13,500
F_2–H_2	0.99	0.97	11.5	0.581	14,100

P_c = 400 psi.
Expansion-area ratio = 30:1.

The differences in propellant fraction for different propellant combinations in a given mission depend primarily on two of the physical properties of the propellant combinations. The temperature range of the liquid phase of the propellant is important as it determines the necessary provisions for insulation, radiant heating or cooling, and/or propellant boiloff that must be made for a particular mission. The propellant bulk density is another important property in that it affects the propellant fraction by changing the weight of tank and associated hardware per pound of propellant.

These two factors are very significant in the design of hydrogen-fueled systems. Liquid hydrogen has the lowest

Table 2. Characteristics of pressurized systems

Propellants	Combustion efficiency	Mixture ratio	Bulk density, g/cc	Effective exhaust velocity, ft/sec
N_2O_4–N_2H_4	0.96	1.25	1.22	9,900
O_2–RP-1	0.95	2.33	0.997	10,050
F_2–N_2H_4	0.96	2.2	1.32	12,280
O_2–H_2	0.965	5.0	0.325	13,280
F_2–H_2	0.965	11.5	0.581	13,880

P_c = 150 psi.
Expansion-area ratio = 20:1.

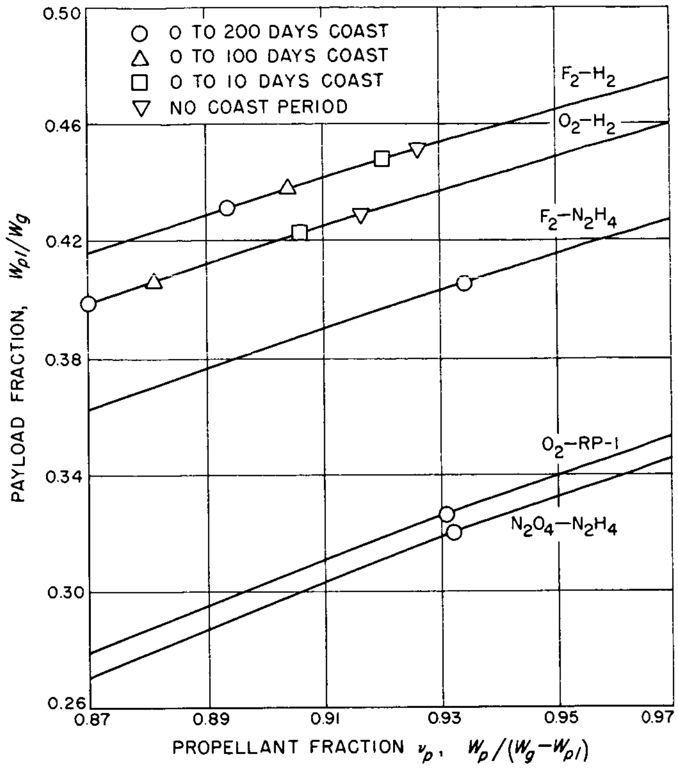

Fig. 1. Relation of payload fraction to propellant fraction for pumped systems with $\Delta V = 10{,}000$ ft per sec

boiling point and the lowest density of any of the proposed fuels. Thus, hydrogen-fueled systems may be expected to have relatively low propellant fractions, particularly for missions involving long storage times. The problem of determining the optimum amount of insulation weight for a particular storage time is discussed in the Appendix.

A smaller, but still significant, effect on the relative propellant fractions comes from a calculation of propellant reserve for specific-impulse fluctuations and guidance errors. In order to provide an identical velocity increment, the high specific-impulse systems must carry a greater percentage of reserve propellant (which is considered as dead weight) than is carried by the low specific-impulse systems according to the relation

$$\frac{\delta\,\Delta V}{\Delta V} = \frac{e^{(\Delta V/c)} - 1}{\Delta V/c}\left(\frac{\delta\,W_p}{W_p}\right) = \text{constant}$$

This results in a smaller value of propellant fraction for high specific-impulse systems as compared with low specific-impulse systems with similar density and temperature characteristics. For example, in a mission with $\Delta V = 20{,}000$ ft per sec and a reserve of 400 ft per sec ($\delta\,\Delta V/\Delta V = 2\%$), a system with specific impulse of 400 sec would require a $\delta W_p/W_p$ of 0.75%, as compared with 0.60% for a system of 300 sec.

The actual magnitude of the variation in propellant fraction with propellant combination depends on several factors of mission and vehicle design. The importance of good liquid-phase temperature range increases with increasing duration of the stage boost and coast periods. In general, those factors which increase the relative proportion of propellant and propellant-dependent weight in a stage make high bulk density more important. This would include missions with high gross velocity or gross-stage weight, and/or vehicle designs with large specific weights of pressurizing gas or insulation.

The relative performance of the different propellant combinations is shown in Fig. 1 through 4. The symbols on the curves represent typical upper-stage designs for the various propellants. The design weights were based on the work of Henneberry, et al (Ref. 1), with the exception that constant values of thrust-to-weight ratio ($\sim$1.0) and gross weight (20,000 lb for pressurized systems and 30,000 lb for pumped systems) were used. These gross weights are believed to be representative of upper

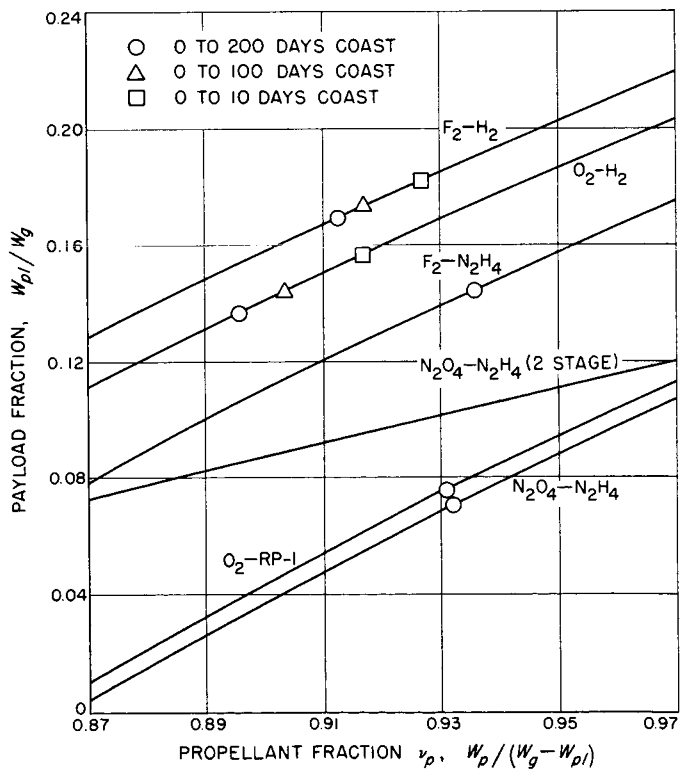

Fig. 2. Relation of payload fraction to propellant fraction for pumped systems with $\Delta V = 20{,}000$ ft per sec

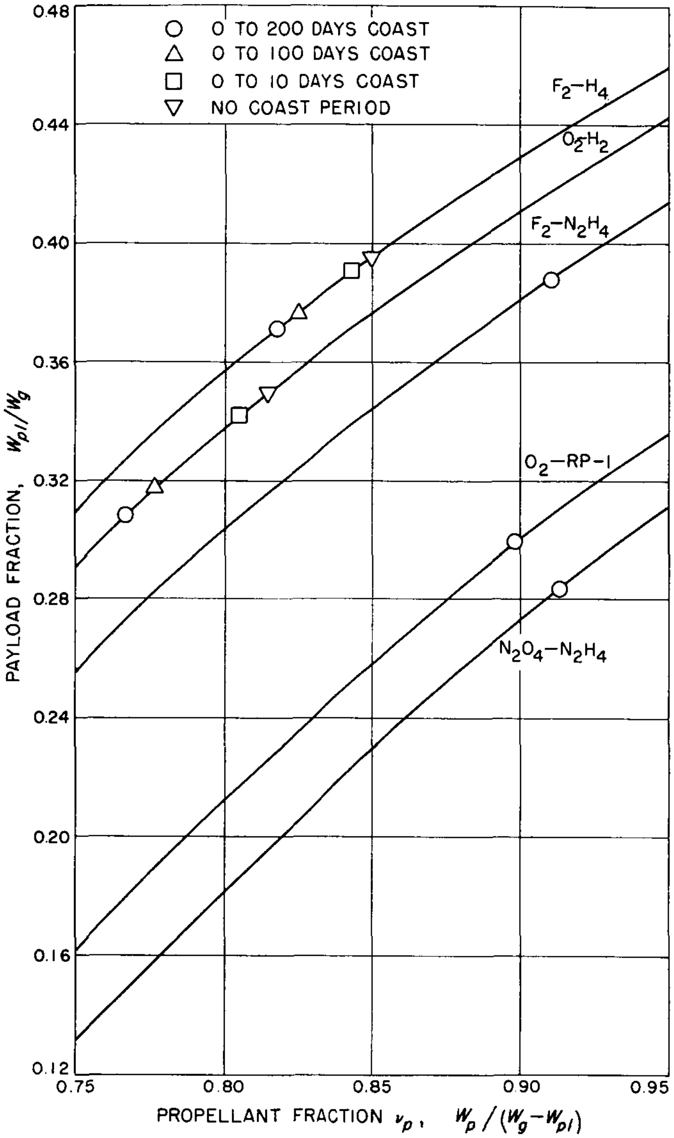

Fig. 3. Relation of payload fraction to propellant fraction for pressurized systems with $\Delta V = 10{,}000$ ft per sec

stages, but the general results should be applicable to stages from a few thousand to nearly 100,000 lb. The hydrogen-fueled systems were modified to reflect the insulation and boiloff requirements for various space-storage times. The actual values of propellant fractions, as shown, might not reflect the ultimate attainable values; however, they are internally consistent inasmuch as they are based on the same design philosophy and, therefore, are believed to offer a representative comparison of the relative values for the different propellant combinations.

The calculation of insulation and boiloff weights (see Appendix) was made with the assumption that the insulation was the Linde SI-4 material (Ref. 3) and that the outer surface was a coating of SiO (Ref. 4). Designs using SI-4 must compensate for the poor structural properties of this material. It should be remembered, however, that considerable structure in the form of aerodynamic shielding or hermetic canisters can be carried during boost, and subsequently jettisoned, with little loss in over-all performance; therefore, the designs have not been penalized.

It is evident from a comparison of the relative costs of a space program that the dominant cost factors are those associated with the operation of the launch site and tracking facilities and procurement and preparation of the payload and the initial booster stage; successive upper stages contribute a less and less significant portion of the over-all program cost. It can be assumed that the fraction of total cost represented by a stage other than the first or second booster could be no greater than 10% and possibly less than 1%, even including all stage-development costs. This relation has two effects on upper-stage development, the most apparent of which is the increased importance of reliability. For instance, if stage reliability could be improved by 5% (either by continued development or by selection of inherently more reliable propellants), the program would be less expensive even if the total stage development costs were to double. The other effect of cost considerations is an increase of emphasis on performance. If it is assumed that additional payload is

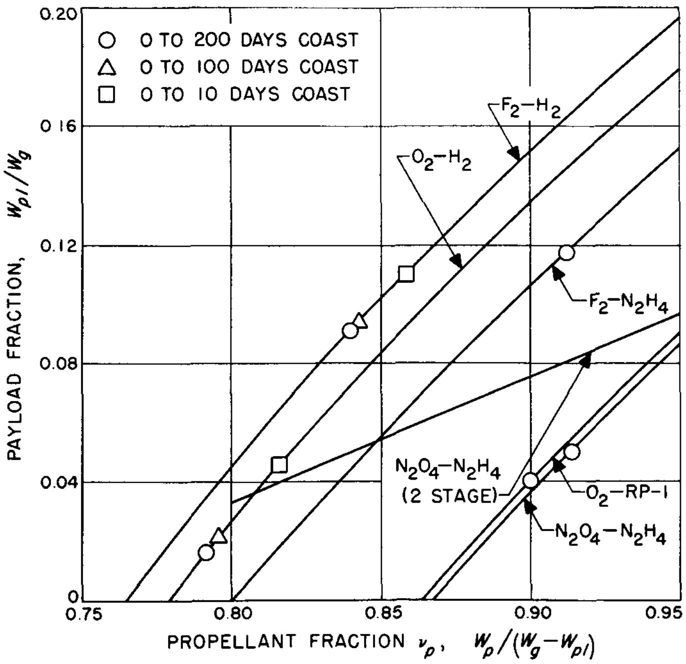

Fig. 4. Relation of payload fraction to propellant fraction for pressurized systems with $\Delta V = 20{,}000$ ft per sec

of the same value to the program as the minimum payload weight, then an upper-stage system which produced 20% more payload is to be preferred over any other, even if the costs of the upper stages should differ by a factor of from 3 to 5. This is approximately the case for F_2–H_2 as compared with O_2–H_2 for pumped systems requiring large ΔV (Fig. 2). For very small payloads, this effect is magnified by considering the proportion of payload weight which is not productive (guidance, structure, etc.). Figure 5 shows to what extent the allowable system cost can increase with increasing performance (or reliability). Any selection, on this basis, of a higher performing but less reliable system assumes the importance of total successful payload weight rather than its distribution among several vehicles. This assumption is probably good for many missions but is unacceptable for others (e.g., manned missions) which emphasize maximum reliability.

Propulsion-system reliability is affected by the propellant combination. For example, freezing of components is a problem in cryogenic systems in general. In addition, fluorine is incompatible with common seal materials; oxygen can absorb nitrogen from the air; and hydrogen is cold enough to condense air. Some other propellants present materials problems of varying degree. At present, attainment of high reliability of the F_2–H_2 combination would seem to require the most effort. Conversely, the N_2O_4–N_2H_4 combination should, ultimately, yield high-reliability systems, possibly surpassing even the highly developed O_2–RP-1 systems.

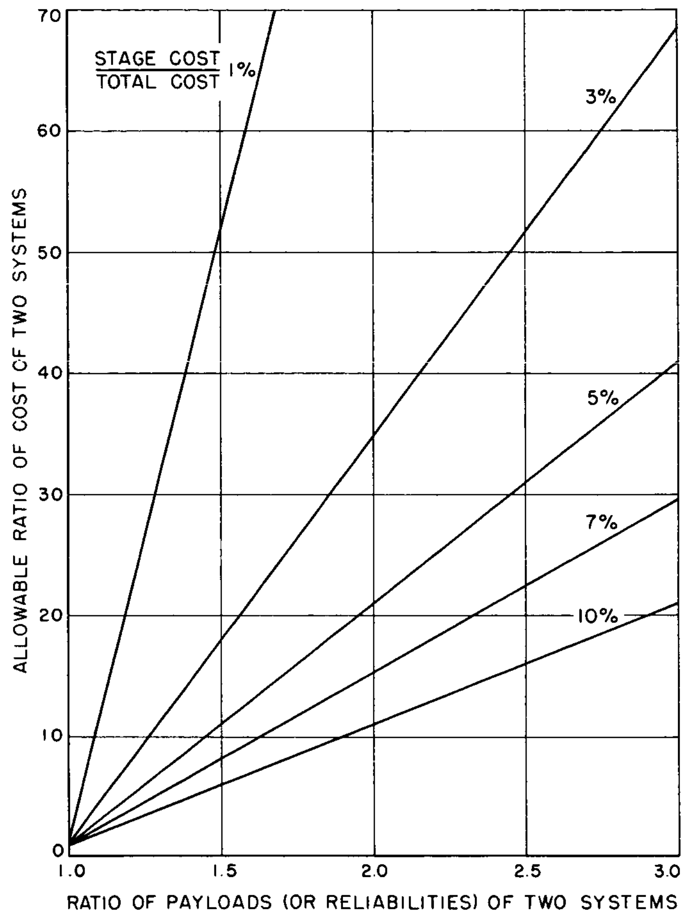

Fig. 5. System cost effects

III. RESULTS AND CONCLUSIONS

Figures 1 and 2 present a comparison of the performance of propellant combinations in pumped systems. It can be seen that the performance ranges of the N_2O_4–N_2H_4 and O_2–RP-1 systems are similar, and that these ranges are from 40 to 80% of the performance of the hydrogen-fueled systems. The F_2–H_2 propellant combination can be expected to have from 5 to 20% better performance than O_2–H_2 systems and up to 25% better performance than F_2–N_2H_4. It is evident, therefore, that the O_2–H_2 and F_2–N_2H_4 systems are comparable in performance, with O_2–H_2 offering a slight advantage ($<8\%$) for all missions but those involving extreme storage times. It would appear, then, that the F_2–H_2 propellant combination would offer considerable performance superiority for all pumped-system applications.

Figures 3 and 4 present a comparison of the performance of propellant combinations in pressurized systems. The performance of F_2–N_2H_4 is almost as good as (for high ΔV, better than) that of the F_2–H_2 systems. The relatively low values of propellant fraction for the O_2–H_2 systems impose a severe performance penalty. The performance of O_2–H_2 systems ranges from 90% to as low as 13% of F_2–N_2H_4 performance. As with pumped systems, the O_2–RP-1 and N_2O_4–N_2H_4 systems give similar performances; that is, from 35 to 80% of F_2–N_2H_4 performance. Thus, F_2–N_2H_4 might prove to be the most versatile propellant combination for pressurized systems. Hydrogen-fueled systems, in general, and O_2–H_2, in particular, are not suited to pressurized-system designs.

The effect of system-cost considerations is such that the performance of the propulsion system is of extreme importance. Thus, because of a potential lower cost of the total system, it is justifiable to expend more effort on the development of the higher-performance propellant combinations (F_2–H_2 and F_2–N_2H_4) than would be necessary for a more simple, lower-performance system.

The importance of propellant bulk density in certain applications introduces the possibility of the increase of bulk density at the expense of specific impulse. This may be accomplished by shifting the system mixture ratio of propellant combinations with dissimilar fuel and oxidizer densities. The potential benefit of this action is greatest in the hydrogen-fueled systems. For example, a 20% density increase could be achieved with F_2–H_2 with a theoretical performance drop of less than 1%. This 20% density increase could result in an increase in propellant fraction of about 1 percentage point, with a resultant net increase in payload capabilities of around 2%. This possibility should be investigated for specific applications.

APPENDIX

Determination of Optimum Insulation Weight

According to Burry and Degner (Ref. 5), it should not be difficult to store propellants of non-cryogenic nature for considerable periods of time anywhere between the orbits of Venus and Mars. The cryogenic oxidizers are slightly more difficult to store, whereas hydrogen requires considerable precautions. If a mission has a requirement for a long storage (or coast) period anywhere within about 5 astronomical units of the Sun, it will be necessary to provide some type of insulation on a hydrogen tank in order to reduce the amount of hydrogen vaporized (and presumed lost—assuming a constant pressure system).

Both the insulation and the lost propellant serve to decrease the value of the stage firing mass ratio, λ, and, therefore, the payload capability of a given system. For a stage with given total tanked propellants and payload, it is desirable to so choose insulation and propellant loss weights that λ is maximized.

The stage firing mass ratio is given by the expression

$$\lambda = \frac{W_g - W_L}{W_g - W_P} \qquad \text{(A-1)}$$

or

$$\lambda = \frac{W_P + W_H + W_I + W_{pl} - W_L}{W_H + W_I + W_{pl}} \qquad \text{(A-2)}$$

where

W_g = stage gross weight at vehicle launch

W_P = total tanked propellant less residuals

W_L = propellant lost by boiloff between vehicle launch and stage firing

W_H = system hardware plus residuals (less insulation)

W_I = tank insulation weight

W_{pl} = stage payload weight (including guidance)

The stage propellant fraction at vehicle launch, ν_p, is given by the expression

$$\nu_p = \frac{W_P}{W_P + W_H + W_I} \qquad \text{(A-3)}$$

Substituting in Eq. (A-2) gives

$$\lambda = \frac{W_P \dfrac{1}{\nu_p} + W_{pl} - W_L}{W_P\left(\dfrac{1}{\nu_p} - 1\right) + W_{pl}} \qquad \text{(A-4)}$$

In order to simplify the determination of the maximum value of λ, it is desirable to eliminate W_L from Eq. (A-4). This can be done by arbitrarily defining an effective propellant fraction ν_p' as follows:

$$\lambda = \frac{W_P \dfrac{1}{\nu_p'} + W_{pl}}{W_P\left(\dfrac{1}{\nu_p'} - 1\right) + W_{pl}} \qquad \text{(A-5)}$$

Combining Eq. (A-2) and Eq. (A-5) and solving for the ν_p' gives

$$\nu_p' = \frac{W_P}{W_P\left(1 + \dfrac{W_H + W_I + W_{pl}}{W_P - W_L}\right) - W_{pl}} \qquad \text{(A-6)}$$

which can be more simply represented by substituting λ again from Eq. (A-2) so that

$$\nu_p' = \frac{W_P}{W_P + W_H + W_I + W_L\left(\dfrac{1}{\lambda - 1}\right)} \qquad \text{(A-7)}$$

The criterion of maximum λ can then be fulfilled by determining the conditions for maximizing ν_p', since, in Eq. (A-5), W_P and W_{pl} are independent of W_L and W_I. It is evident that proper design of the stage would allow any required value of W_L to be provided without changing the engine mixture ratio merely by allowing the tanked mixture ratio to differ from the engine mixture ratio. Therefore, it is desirable to so select W_I and W_L that ν_p' is maximized. Using Eq. (A-6) and assuming that $\partial W_H/\partial W_L$ and $\partial W_H/\partial W_I$ are small (which is a good first-order assumption), the relation for maximum ν_p' simplifies to

$$W_I = W_L\left(\frac{1}{\lambda - 1}\right) \qquad \text{(A-8)}$$

Using this relation in Eq. (A-7) yields

$$\nu_p' = \frac{W_P}{W_P + W_H + 2W_I} \qquad \text{(A-9)}$$

In hydrogen-fueled systems, the hydrogen tank presents the larger volume as well as the most severe insulation requirement. Therefore, the insulation weights for the system can be closely approximated by calculating the requirement for the hydrogen tank alone. Assuming a heat balance between the Sun and the night sky, Roschke (Ref. 3) gives, for an isolated spherical tank with uniform liquid temperature,

$$\frac{\varepsilon}{\alpha}\sigma T_2^4 + \frac{\bar{k}}{\alpha(\Delta x)}(T_2 - T_L) = \frac{C_S}{4R^2} \tag{A-10}$$

and

$$q_{2L} = 4\pi a^2 \frac{\bar{k}}{(\Delta x)}(T_2 - T_L) \tag{A-11}$$

where

ε = infrared emissivity of outer surface

α = solar absorptivity of outer surface

σ = Boltzmann constant

T_2 = temperature of outer surface

T_L = temperature of tank wall (and liquid)

$\bar{k}$ = average thermal conductivity of insulation

Δx = thickness of insulation

R = distance to center of Sun, A.U.

C_S = solar intensity at $R = 1$

a = tank radius

q_{2L} = net heat flow to liquid

Changing parameters and remembering Eq. (A-8) gives

$$\frac{\varepsilon}{\alpha}\sigma\left(\frac{H_v W_L^2}{16\pi^2 a^4 \bar{k} \rho_I \Delta t(\lambda - 1)} + T_L\right)^4 + \frac{H_v W_L}{4\pi a^2 \alpha \Delta t} = \frac{C_S}{4R^2} \tag{A-12}$$

where

H_v = heat of vaporization of liquid

ρ_I = density of insulation

Δt = storage (coast) period

Equation (A-12) can be solved numerically for the optimum value of W_L for a given set of conditions. From Eq. (A-8), the optimum value of W_I is obtained.

NOMENCLATURE

a tank radius

c engine effective exhaust velocity

C_S solar intensity at $R = 1$

H_v heat of vaporization of liquid

$\bar{k}$ average thermal conductivity of insulation

P_c chamber pressure

q_{2L} net heat flow to liquid

R distance to center of Sun, A. U.

T_2 temperature of outer surface

T_L temperature of tank wall (and liquid)

W_g stage gross weight at launch

W_H system hardware plus residuals (less insulation)

W_I tank insulation weight

W_L propellant lost by boiloff between launch and firing

W_p stage useful-propellant weight

W_P total tanked propellant less residuals

W_{pl} stage payload weight (including guidance)

α solar absorptivity of outer surface

Δt storage (coast) period

ΔV stage gross (or field-free) velocity

Δx thickness of insulation

ε infrared emissivity of outer surface

λ stage firing mass ratio, exp $(\Delta V/c)$

ν_p propellant fraction at launch, $W_p/(W_g - W_{pl})$

ρ_I density of insulation

σ Boltzmann constant

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